NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

REPORT No. 403

ICE PREVENTION ON AIRCRAFT BY MEANS OF ENGINE EXHAUST HEAT AND A TECHNICAL STUDY OF HEAT TRANSMISSION FROM A CLARK Y AIRFOIL

By THEODORE THEODORSEN and WILLIAM C. CLAY

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AERONAUTICAL SYMBOLS

1. FUNDAMENTAL AND DERIVED UNITS

	Symbol	Metric		English	
		Unit	Symbol	Unit	Symbol
Length	l	meter	m	foot (or mile)	ft. (or mi.)
Time	t	second	s	second (or hour)	sec. (or hr.)
Force	F	weight of one kilogram	kg	weight of one pound	lb.
Power	P	kg/m/s		horsepower	hp
Speed		{km/h	k. p. h.	mi./hr.	m. p. h.
		{m/s	m. p. s.	ft./sec.	f. p. s.

2. GENERAL SYMBOLS, ETC.

W, Weight $=mg$

g, Standard acceleration of gravity $=9.80665$ m/s^2 $=32.1740$ ft./sec.^{2}

m, Mass $=\frac{W}{g}$

ρ, Density (mass per unit volume).

Standard density of dry air, 0.12497 (kg-m^{-4} s^2) at 15° C. and 760 mm $=0.002378$ (lb.-ft.$^{-4}$ sec.^{2}).

Specific weight of "standard" air, 1.2255 kg/m^3 $=0.07651$ lb./ft.^{3}.

mk^2, Moment of inertia (indicate axis of the radius of gyration k, by proper subscript).

S, Area.

S_w, Wing area, etc.

G, Gap.

b, Span.

c, Chord.

$\frac{b^2}{S}$, Aspect ratio.

μ, Coefficient of viscosity.

3. AERODYNAMICAL SYMBOLS

V, True air speed.

q, Dynamic (or impact) pressure $=\frac{1}{2}\rho V^2$.

L, Lift, absolute coefficient $C_L=\frac{L}{qS}$

D, Drag, absolute coefficient $C_D=\frac{D}{qS}$

D_o, Profile drag, absolute coefficient $C_{D_o}=\frac{D_o}{qS}$

D_i, Induced drag, absolute coefficient $C_{D_i}=\frac{D_i}{qS}$

D_p, Parasite drag, absolute coefficient $C_{D_p}=\frac{D_p}{qS}$

C, Cross-wind force, absolute coefficient $C_C=\frac{C}{qS}$

R, Resultant force.

i_w, Angle of setting of wings (relative to thrust line).

i_t, Angle of stabilizer setting (relative to thrust line).

Q, Resultant moment.

Ω, Resultant angular velocity.

$\rho\frac{Vl}{\mu}$, Reynolds Number, where l is a linear dimension.

e. g., for a model airfoil 3 in. chord, 100 mi./hr. normal pressure, at 15° C., the corresponding number is 234,000;

or for a model of 10 cm chord 40 m/s, the corresponding number is 274,000.

C_p, Center of pressure coefficient (ratio of distance of $c. p.$ from leading edge to chord length).

α, Angle of attack.

ϵ, Angle of downwash.

α_o, Angle of attack, infinite aspect ratio.

α_i, Angle of attack, induced.

α_a, Angle of attack, absolute.
(Measured from zero lift position.)

γ Flight path angle.

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By THEODORE THEODORSEN and WILLIAM C. CLAY
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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

NAVY BUILDING, WASHINGTON, D. C.

(An independent Government establishment, created by act of Congress approved March 3, 1915, for the supervision and direction of the scientific study of the problems of flight. Its membership was increased to 15 by act approved March 2, 1929 (Public, No. 908, 70th Congress). It consists of members who are appointed by the President, all of whom serve as such without compensation.)

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SUMMARY

This investigation was conducted by the National Advisory Committee for Aeronautics to study the practicability of employing heat as a means of preventing the formation of ice on airplane wings. The report relates essentially to technical problems regarding the extraction of heat from the exhaust gases and its proper distribution over the exposed surfaces. In this connection a separate study has been made to determine the variation of the coefficient of heat transmission along the chord of a Clark Y airfoil.

A study of the heat transmission from the airfoil reveals that the local transmission is very high at the leading edge and that it decreases rapidly to a minimum value at a point located about 30 per cent back along the chord.

Experiments on ice prevention both in the laboratory and in flight show conclusively that it is necessary to heat only the front portion of the wing surface to effect complete prevention.

The marked variation in the heat transmission over the front portion of the wing makes the problem of an efficient heat-distributing system a matter of some technical difficulty. The actual quantity of heat needed for ice prevention is, however, surprisingly small, being in the order of one-tenth of that available in the engine exhaust gases.

The relative merits of various methods of ice prevention by means of heat are analyzed with the result that a vapor system is found to offer the most satisfactory solution, especially for airplanes which are not constructed entirely of metal. In "all-metal" designs it may be entirely practicable to employ a direct exhaust-heating system.

Experiments in flight show that a vapor-heating system which extracts heat from the exhaust and distributes it to the wings is an entirely practical and efficient method for preventing ice formation.

A narrow slot on the upper surface located about one-tenth of the chord length from the leading edge is employed in these tests for the purpose of collecting the water which would otherwise blow back and freeze aft of the heated leading edge. The tests seem to indicate, however, that this slot may not be essential.

INTRODUCTION

Formations of ice occurring on aircraft are most prominent and detrimental on the wing surfaces. Accumulations on the propeller are probably almost always concentrated near the hub and are generally small and cause little trouble (except in cases of airships where small pieces of ice are thrown into the envelope). Some ice, of course, forms on wires, struts, tail surfaces, and other appendages; on airplanes which have several bays with numerous members in the air stream creating a large parasite drag, this item is likely to become of some consequence. Modern design, however, tends toward the removal of these extraneous structures, and it is generally conceded that the elimination of ice formation on the wings will practically nullify the factors which usually cause a forced landing.

The various factors causing ice formations on airplanes in flight have been discussed in earlier reports (references 3 and 4) and will not be repeated here. However, some points may be brought out which have a bearing on this particular investigation. Experience has shown that there exist two extreme cases of ice formation apparently differing widely in nature. The formation caused by comparatively large drops in an air temperature slightly below the freezing point is known as *glaze* ice. It occurs frequently and builds up rapidly in the form of an irregular transparent coating. At temperatures much below freezing the existing drops are usually much smaller and greatly undercooled. They freeze instantly on hitting the airfoil and form a whitish regular coating called *rime*, which builds up slower than the glaze ice and has a less detrimental effect on the aerodynamic characteristics of the airfoil.

Experiments conducted under reference 4 showed that ice accumulated only at or near the leading edge. Consequently, the objects of the present investigation were to determine the practicability of heating only the front portion of the wing and to determine the quantity of heat needed for this purpose.

The utilization of engine-exhaust heat as a means of preventing the formation of ice on aircraft presents a possibility that has been under discussion for some time. Several other methods have been investigated

as possibilities in the last few years, but most of them have merely resulted in establishing the conclusion that there exists no simple means of preventing ice formation on airplane parts. The use of engine-exhaust heat has naturally been about the last method to be studied, because of the necessary complications in design, construction, and operation which such a method suggests.

The design of a practical heating system which will maintain the exposed surface of a wing at a sufficient temperature to prevent or remove ice formation entails a study of the convection of heat from an airfoil. Some investigations of this nature have already been conducted by others both in regard to ice prevention and in connection with wing-radiator systems. (References 1, 2, and 3.)

In reference 2 are many curves obtained from tests on an R. A. F. 26 wing section to determine the distribution of heat transmission around the entire surface. The test model was divided into a great many sections, and thereby a very detailed representation of the heat transmission was obtained. The results show conclusively that the transmission of heat at the tip of the leading edge is very high compared to the rest of the surface. As the R. A. F. 26 section is very thin, these results are not directly applicable to the comparatively thick sections which are more widely used in this country. In addition, the model used was small, making the extrapolation of the results to full scale uncertain.

In reference 3 are given results of an investigation of the transmission of heat from several small strut sections tested in a wind tunnel under ice-forming conditions. Although obtained at very low Reynolds Numbers, the results seem to indicate that there is sufficient heat in the exhaust of the average engine to keep ice off the wings. The tests were made primarily to estimate the heat required to keep the entire surface of a wing sufficiently heated to prevent the formation of ice thereon. From a practical viewpoint, however, it is obvious that any design which would provide for heating the entire wing, regardless of method, would not only be excessively heavy, but would also involve such radical changes in wing construction that its adaptation to present aircraft would be very difficult, and it is doubtful if the benefits derived therefrom would be considered as justifiable.

This report is divided into two parts. Part I is essentially a theoretical discussion of the question of heat transfer in general, and also includes an experimental study of variations in the heat transfer coefficients along the chord of a Clark Y airfoil. Such a study is not only of assistance in the development of apparatus for ice prevention, but may be of interest in connection with wing radiators and in other studies that are concerned with the flow of air about an airfoil.

Part II deals with the extraction and distribution of engine exhaust heat necessary to prevent the accumulation of ice on the leading edge of an airplane wing, together with the study and design of a suitable collector to remove the water drops as they are blown back from the heated area.

This investigation was made by the National Advisory Committee for Aeronautics at Langley Field, Va.

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PART I

HEAT TRANSMISSION ON AIRFOILS

Theoretical discussion of the heat transmission.—In a liquid flowing at a high velocity past a hot body which is completely immersed there exists a thin layer along the surface of the body in which the temperature gradient normal to the surface is rather great. Outside of this boundary layer the flow of energy from one particle to another is entirely negligible.

It appears that Oberbeck (reference 5) was the first investigator to analyze this problem. He combined equations of motion of the fluid with the differential equations of the heat flow. His results are, however, only applicable in certain simple cases.

The similarity between the problem of heat transmission and the problem of fluid friction presents itself at once. It is known from aerodynamic considerations that the flows are similar if the Reynolds Numbers are equal. If the fluid flows are similar we must obviously expect a certain similarity also in the heat flows.

A rational basis for work on heat transmission (reference 6) was created by Nusselt. He showed that by considering solely the *dimensions* of the differential equations it was possible to obtain very important and useful information. He found that it is unnecessary to consider the effect on the heat transmission of *each* of the variable factors; his analysis showed that such factors always would appear in certain nondimensional combinations or "Kenngroessen." If the effect of one of these factors is measured the effect of the remaining factors of the group can be predicted. Nusselt confirmed his analysis by numerous tests.

The case is particularly simple for diatomic gases. The heat transmission coefficient for a body in an air stream flowing at a certain angle relative to the body is given by the formula (reference 7)

$$\alpha=\frac{\lambda_m}{l}\psi\left(\frac{lV_o\,\rho_m}{\mu_m}\right) \qquad \text{(I)}$$

where

α—the heat transmission coefficient
λ_m—the conductivity of the air in the middle of the temperature range
l—a representative dimension
ρ_m—the average density of the medium
μ_m—the average viscosity of the medium
V_o—the velocity of the fluid at a great distance from the body.

The form of the function ψ must, however, be found by experiment for each type of body and for each direction of the flow. We are mostly concerned with two-dimensional flow. In this case the variables are the cross section of the body and the angle of some representative line measured relative to the direction of flow.

The simplest case is that of a circular cylinder because no direction relative to the two-dimensional flow need be specified. Nusselt found from a series of tests by Hughes the equation

$$\alpha=0.067\frac{\lambda_m}{d}\left(1273+\frac{dV_o\,\rho_m}{\mu_m}\right)^{0.716} \qquad \text{(II)}$$

where d is the diameter of the tube. This value of α represents an average for the entire cylinder. The transmission on the front of the cylinder is actually much greater than that on the rear.

The question of the local heat transmission is rather essential in connection with the subject matter of this report. It is to be expected that the heat transmission for a particular point on a complex body is quite different from the average value.

In the testing of airfoils there exists no known relation between the values of the heat transmission for the various angles of attack. All that can be concluded from the theory of Nusselt is that the heat transmission in each case is a (different) function of the Reynolds Number; the airfoil at various angles of attack must in each case be considered as a different body. Consequently measurements must be made at several angles of attack.

As the velocity of flow near the leading edge is somewhat greater than the velocity of translation, and as the boundary layer here is thin, the rate of heat transfer in this region is greater than that for the remainder of the airfoil.

We will also, at this point, call attention to a condition which is of particular interest in connection with the question of ice formation. There occurs an adiabatic expansion of the air flowing along the upper surface of an airfoil, which results in a considerable temperature drop. This drop is expressed by the relation

$$\frac{dT}{dp}=\frac{k-1}{k}\,\frac{T}{p}$$

If the pressure drop corresponds to q, which at 100 miles per hour is about 5 inches of water, we obtain for conditions at the freezing point

$$dT=\frac{5}{380}\times\frac{0.4}{1.4}\times460=1.73^{\circ}\text{ F.}$$

This temperature drop is not at all negligible. It has been shown (reference 8) that a pressure difference of approximately $24q$ per radian, with the angle measured from the ideal angle of attack, occurs between the upper and lower surface of the average airfoil at a point located very close to the leading edge. The ideal angle of attack is defined as the angle at which no pressure difference exists at the leading edge.

If the airplane is operating at but 2° beyond the ideal angle of attack, this pressure difference amounts to $0.85q$ at 100 miles per hour. The adiabatic temperature drop on the upper surface is thus of the order of 1° F. This temperature drop thus occurs at *normal* conditions. As the airplane becomes loaded with ice the angle of attack must be increased accordingly, resulting in a rapidly increasing temperature drop. It is worth notice that the lowest temperature occurs where the velocity is greatest, which means that the greatest temperature difference occurs at the very point where the rate of heat transfer is the greatest. The boundary layer just above the center of curvature is, in addition, exceedingly thin.

It is known that dangerous ice formations occur most frequently at temperatures very near the freezing point. With the air temperature only slightly below freezing the condition mentioned above becomes doubly important. The differential temperature at the point referred to may be very great compared with that of the remainder of the airfoil.

It has been shown in earlier experiments (reference 4) that ice usually forms on the upper surface near the leading edge. As soon as the formation is started its poor aerodynamic properties will give rise to excessive local velocities. There is a tendency to extend the formation in a rearward direction probably owing to a greater velocity and a resulting lower temperature existing just behind the peaks of the projecting ridges.

Apparatus and methods.—To determine the quantity of heat necessary to maintain the surface of an airplane wing at a temperature sufficient to prevent the formation of ice thereon and to obtain general information about the distribution of the heat transmission around the surface of an airfoil, tests were made on a heated model in a wind tunnel.

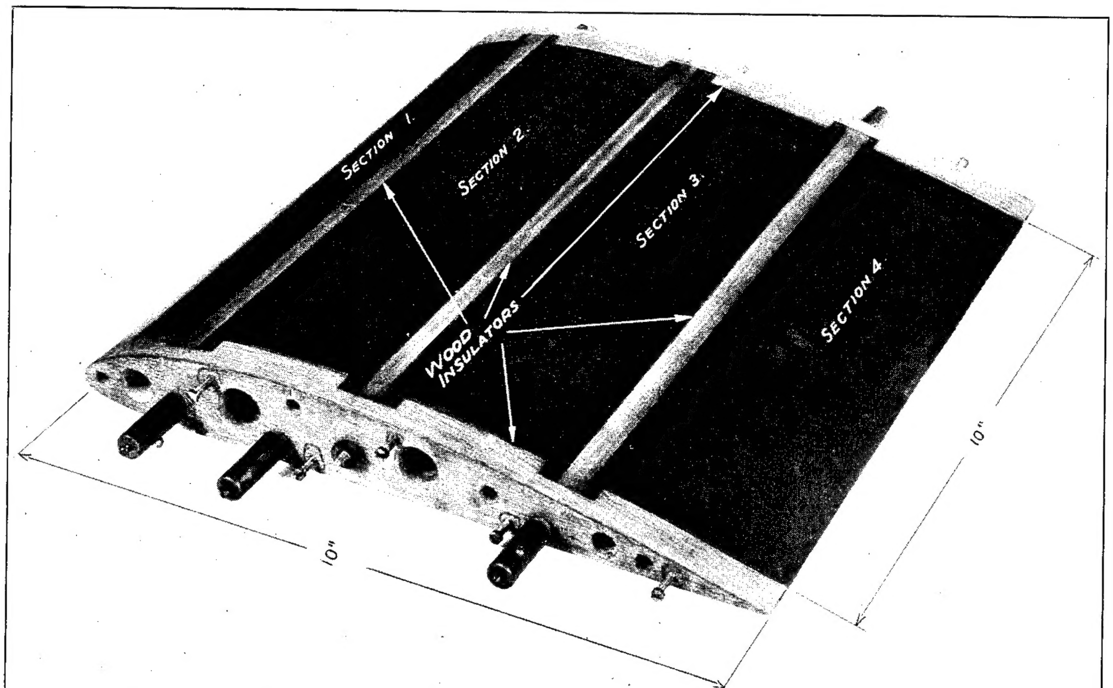

FIGURE 1.—Model for experiments on heat transmission

It was not considered necessary to make a point-to-point survey of the heat transmission, as we were interested principally in the total heat required for a certain portion of the leading edge. Such tests, however, would probably be of considerable theoretical interest in connection with the effect of the leading-edge curvature on the maximum rate of heat transmission and its relation to the airfoil efficiency.

For these tests an electrically heated, solid brass wing (fig. 1) was constructed in four separate sections for the purpose of measuring the quantity of heat necessary to maintain a certain temperature difference between the surface of each section and the air stream.

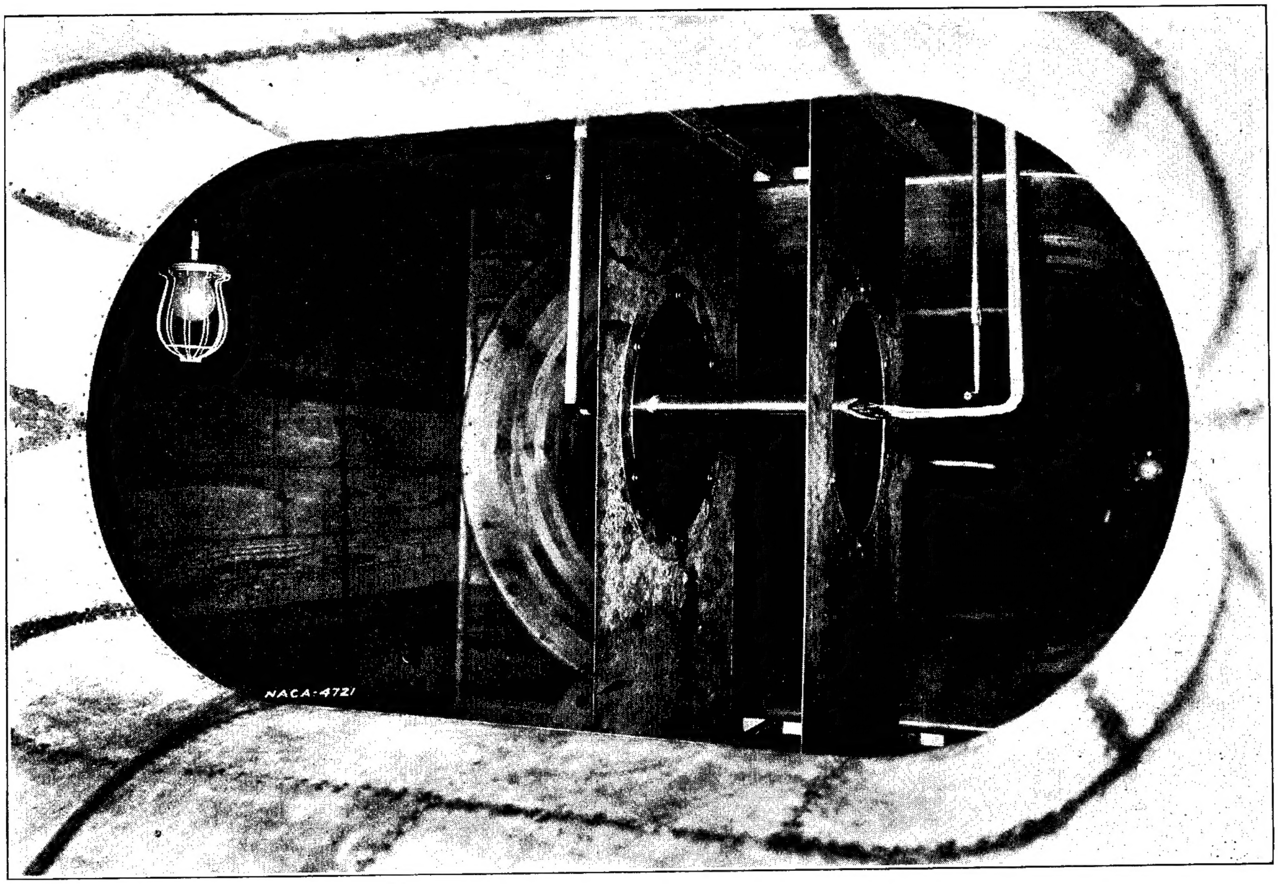

FIGURE 2.—Experimental airfoil mounted in wind tunnel

These sections were separated by wooden strips to minimize conduction from one to the other. A Clark Y section was chosen for these tests as being fairly representative of most wing sections used in this country. The wing had a span of 11½ inches and chord of 10 inches. It was supported in the air stream between large end plates (fig. 2), employed to approach the condition of two-dimensional flow.

The leading edge section of the airfoil (section 1) was designed to include about 12 per cent of the chord. It was found from earlier experiments that this is as far back as ice forms and is therefore the part about which information on heating is of particular importance. The other sections were arranged as shown in the photograph in Figure 1 and in the diagram in Figure 15. Two holes were drilled through each section in the direction of the span: a large one to accommodate an electric heating element and a small one to accommodate a thermocouple.

Each heating element consisted of Nichrome wire wound around a small Alundum rod. The elements were packed at both ends with asbestos to support them centrally and to reduce heat loss. The wires supplying current to these elements were led out of the air stream through hollow struts as shown in Figure 2. The power was taken from a 110-volt direct-current source and the input to each section controlled by a separate rheostat. A wiring diagram of the heat-control apparatus is shown in Figure 3, and

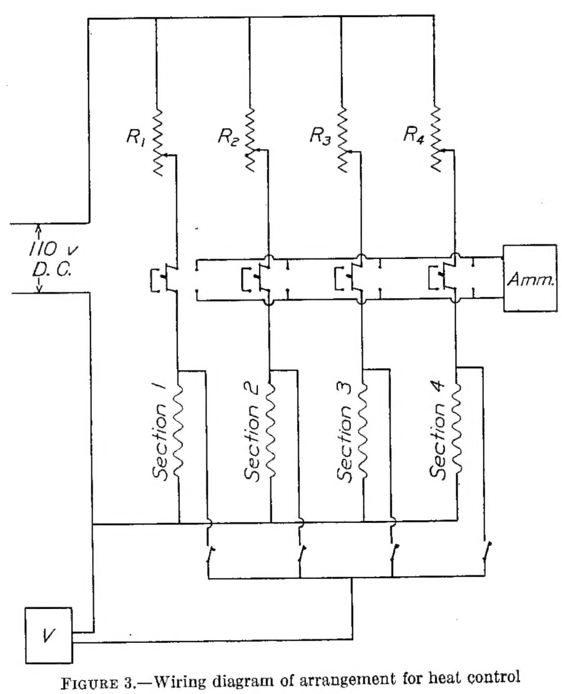

FIGURE 3.—Wiring diagram of arrangement for heat control

a photograph of the entire external installation is shown in Figure 4.

A copper-constantan thermocouple unit was placed in the small hole in each section, so that the hot junction was at a point approximately at the midspan of the airfoil. The cold junctions were placed in the free air stream about 6 inches away from the end plates. The four thermocouples were made of heavy wires and connected to a millivoltmeter by mercury switches to reduce as far as possible the total resistance of the circuit.

The tests were made in the model of the full-scale wind tunnel at the Langley Memorial Aeronautical Laboratory. This tunnel has a jet 2 by 4 feet and a maximum air velocity of about 82 miles per hour with a velocity variation across the test section of less than 1½ per cent. A Pitot tube mounted at a single station was used to indicate the air velocity.

The essential requirements in this study of heat convection are: first, the maintenance of a uniform temperature of the airfoil surface; second, the maintenance and measurement of a definite temperature difference between the airfoil surface and the air stream; and third, the measurement of the energy output due to heat dissipation from the airfoil.

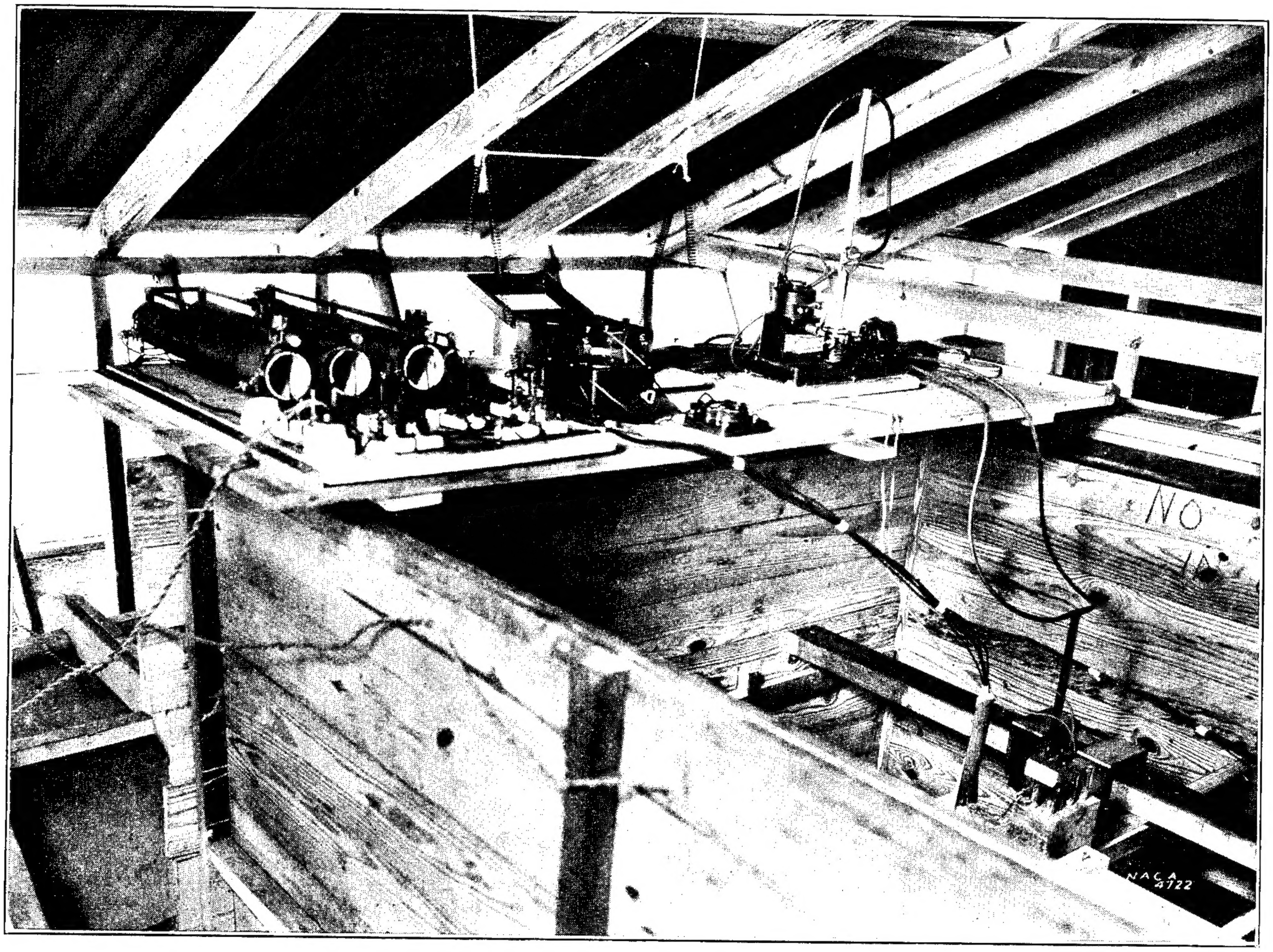

FIGURE 4.—Control group of wind tunnel set-up

For purposes of providing a check on the accuracy of the results and for comparison, tests were also made on a brass cylinder which had a surface area approximately equal to that of the front section of the airfoil. This cylinder was thirteen-sixteenths inch outside diameter and had a wall thickness of one-eighth inch. It was mounted between the end plates and heated in a manner similar to that of the airfoil. The hot junction of the thermocouple in this case was attached directly to the outside surface of the cylinder. Aside from this, the cylinder set-up was identical with that of the airfoil.

The maintenance of a uniform temperature over the surface of each section (fig. 1) was accomplished by making each section of solid metal. The conductivity of heat throughout the entire mass of each section was so great compared to the convection of heat from the surface that the temperature gradient throughout the mass was negligible. For this reason a thermocouple placed practically anywhere inside the section will record satisfactorily the temperature of the surface of that section.

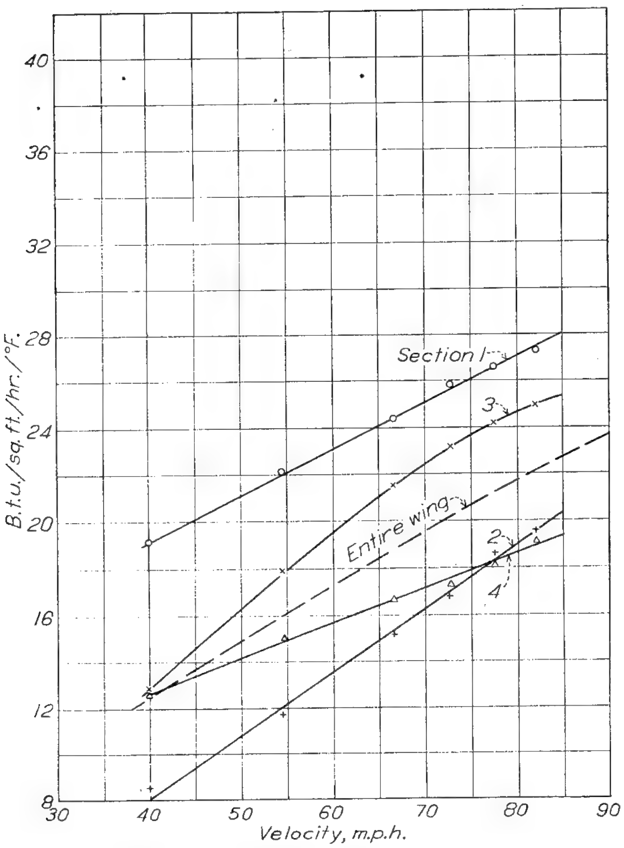

FIGURE 5.—Rate of heat transmission from airfoil. Chord=10 in. Angle of attack=0°

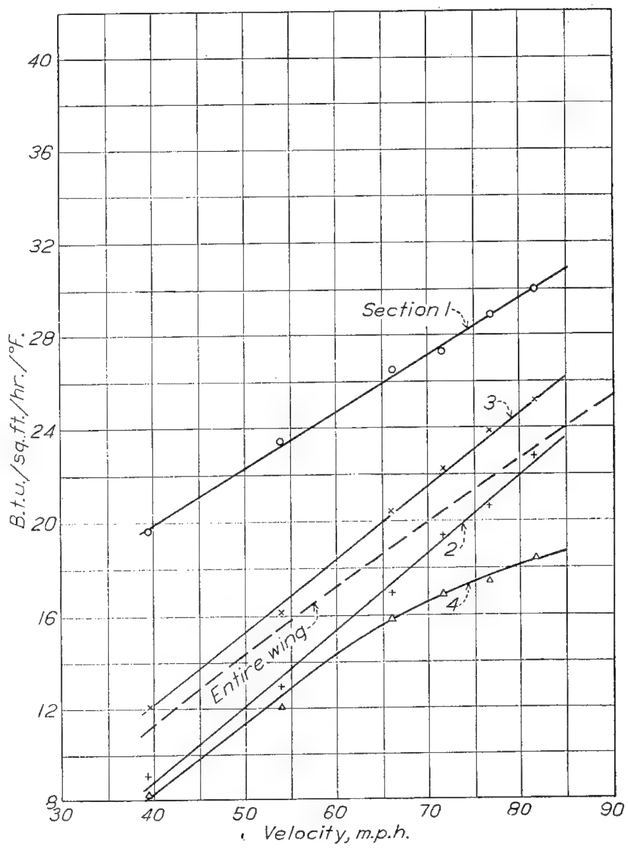

FIGURE 6.—Rate of heat transmission from airfoil Chord=10 in. Angle of attack=6°

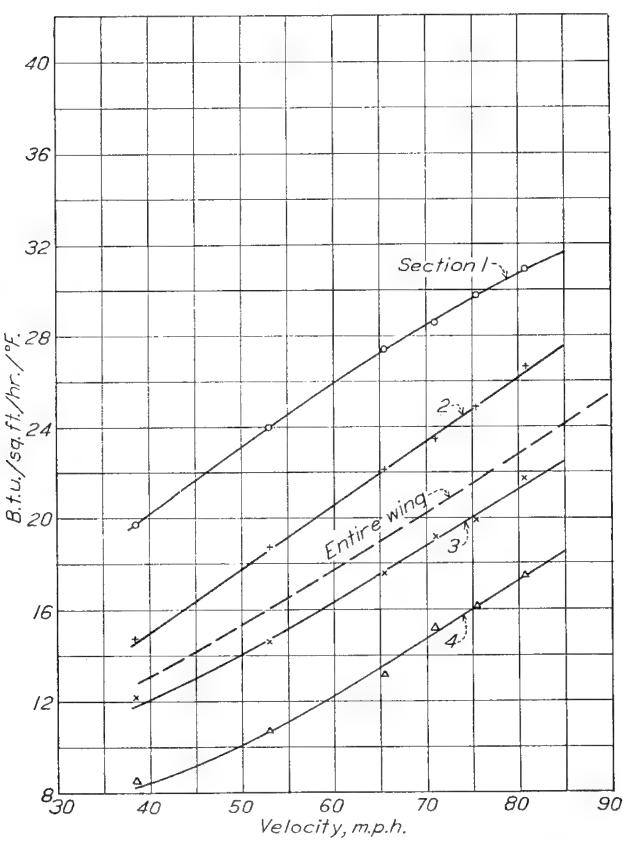

FIGURE 7.—Rate of heat transmission from airfoil. Chord=10 in. Angle of attack=12°

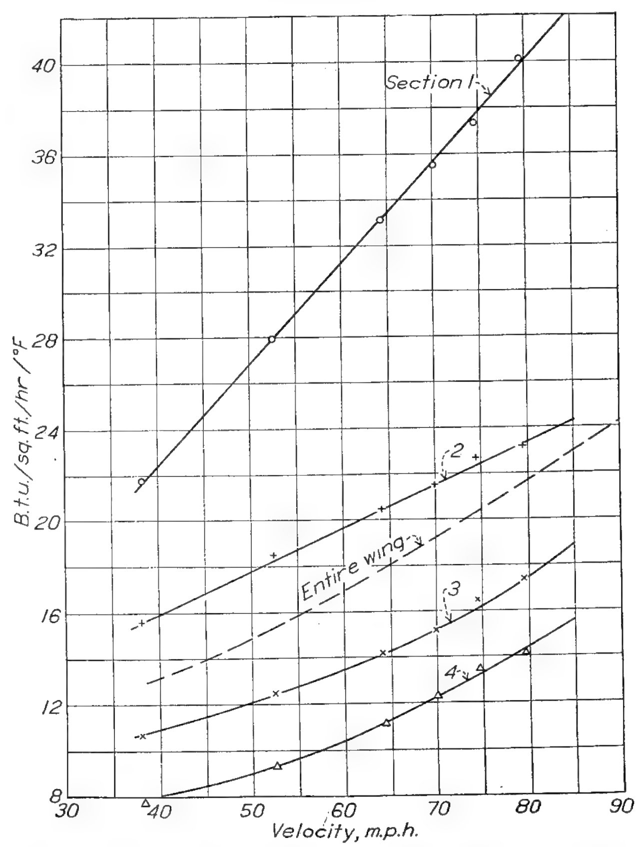

FIGURE 8.—Rate of heat transmission from airfoil. Chord=10 in. Angle of attack=18°

The use of thermocouples as a method of temperature measurement was in this case particularly advantageous, and it was found possible by the use of a sensitive millivoltmeter to keep the errors of measurement within ±0.7° F. By employing a working temperature of about 54° F. above that of the air stream the error of measurement was less than 1.3 per cent.

As is the case of most heat transmission measurements, considerable time was required to obtain temperature equilibrium. The heat absorbed by the capacity is quite considerable, and appreciable inaccuracy may result if the condition of temperature equilibrium is not attained. Errors due to this cause limited the actual test accuracy to about 3 per cent.

Transfer of heat between the individual sections was prevented by maintaining all sections at the same temperature. The heat loss through the wooden end pieces to the end plates was negligible. Hence the measured input in watts to each section was considered to be emitted by that section to the air stream alone.

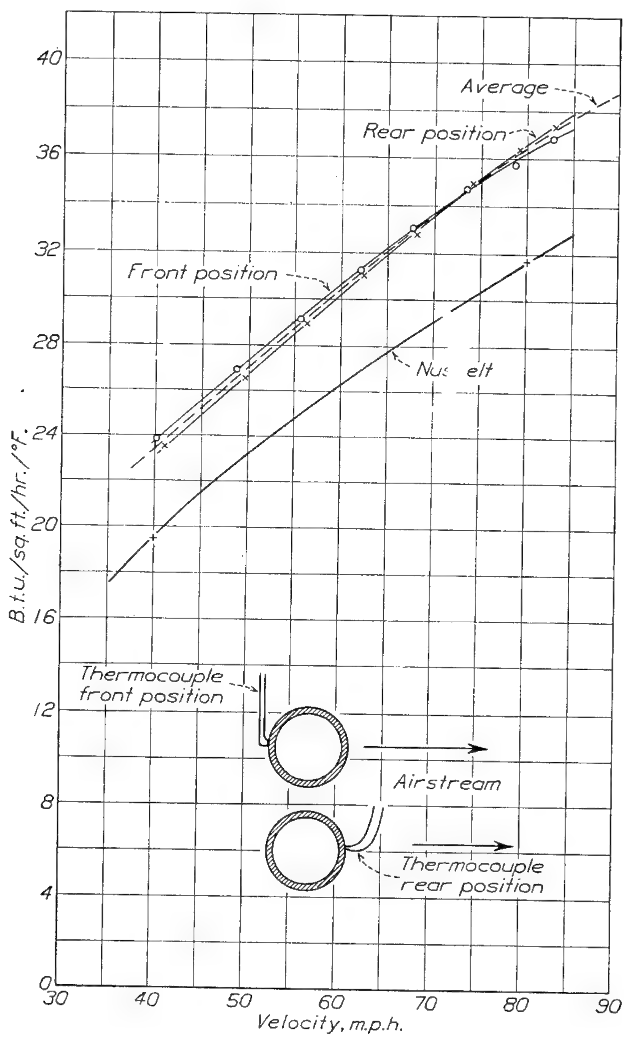

FIGURE 9.—Rate of heat transmission for cylinder (13/16 in. O. D.). Lower curve shows values calculated from formula by Nusselt

Tests on the cylinder were made with the thermocouple located in a front as well as in a rear position to disclose any temperature difference due to insufficient conductivity of the walls.

Tests and results.—During all tests the temperature of each of the four sections of the wing model and of the cylinder was held to a constant value of very nearly 54° F. above that of the air stream. This temperature was low enough to justify neglecting the radiation and high enough to obtain sufficient accuracy.

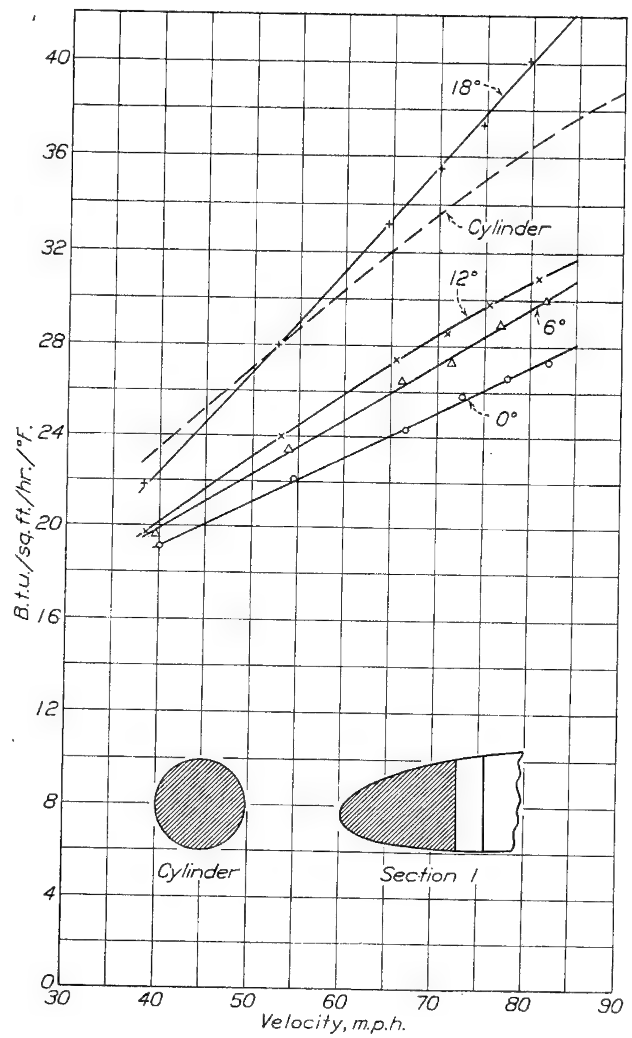

FIGURE 10.—Rate of heat transmission for section 1 of airfoil and the cylinder (the surface area of both are equal)

Tests were made at velocities varying between 40 and 80 miles per hour for four angles of attack; namely, 0°, 6°, 12°, and 18°.

The data from these tests are given in graphical form. (Figs. 5 to 8.) The input (or output) as measured in watts was converted to British thermal units per hour per square foot per Fahrenheit degree, and represents the convection of heat from the airfoil surface to the air stream.

In Figure 9 the heat transmission is plotted against air velocity for the cylinder. One curve is shown for the front and one for the rear location of the thermocouple. The average curve shown is used for further comparisons.

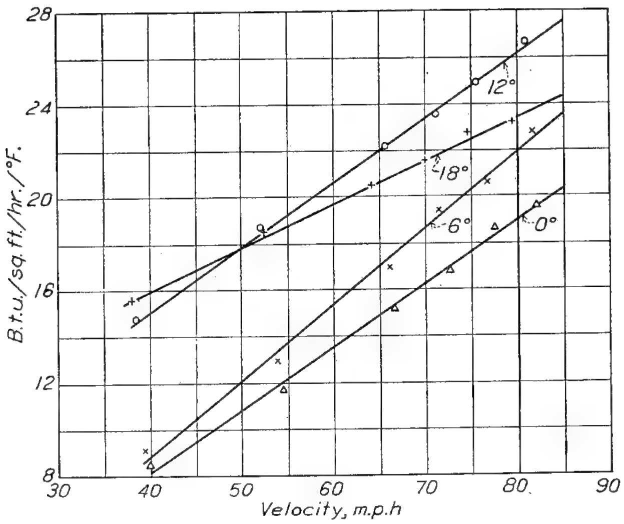

FIGURE 11.—Rate of heat transmission from the airfoil for section 2

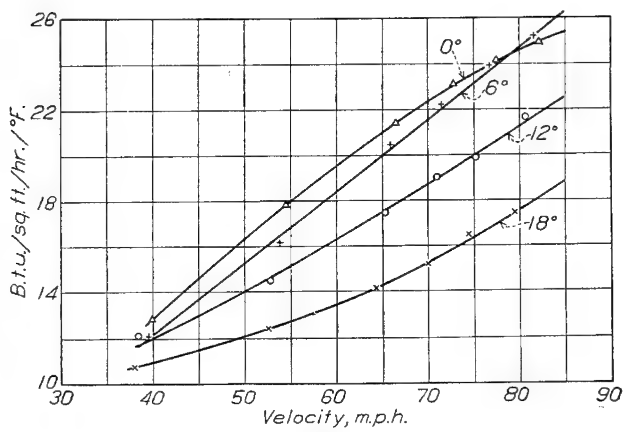

FIGURE 12.—Rate of heat transmission from the airfoil for section 3

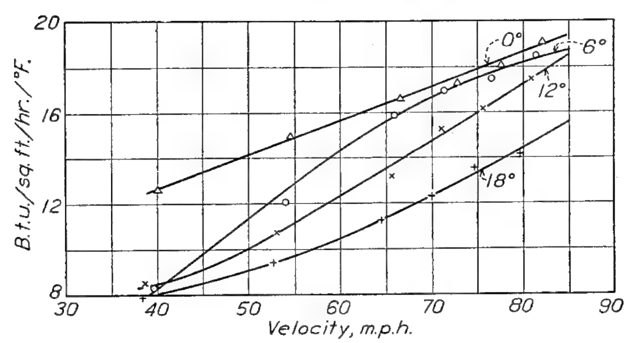

FIGURE 13.—Rate of heat transmission from the airfoil for section 4

section, as mentioned, is of main importance as regards ice formation.

Figure 14 shows a plot of curves for both the airfoil as a whole at different angles of attack and for the cylinder.

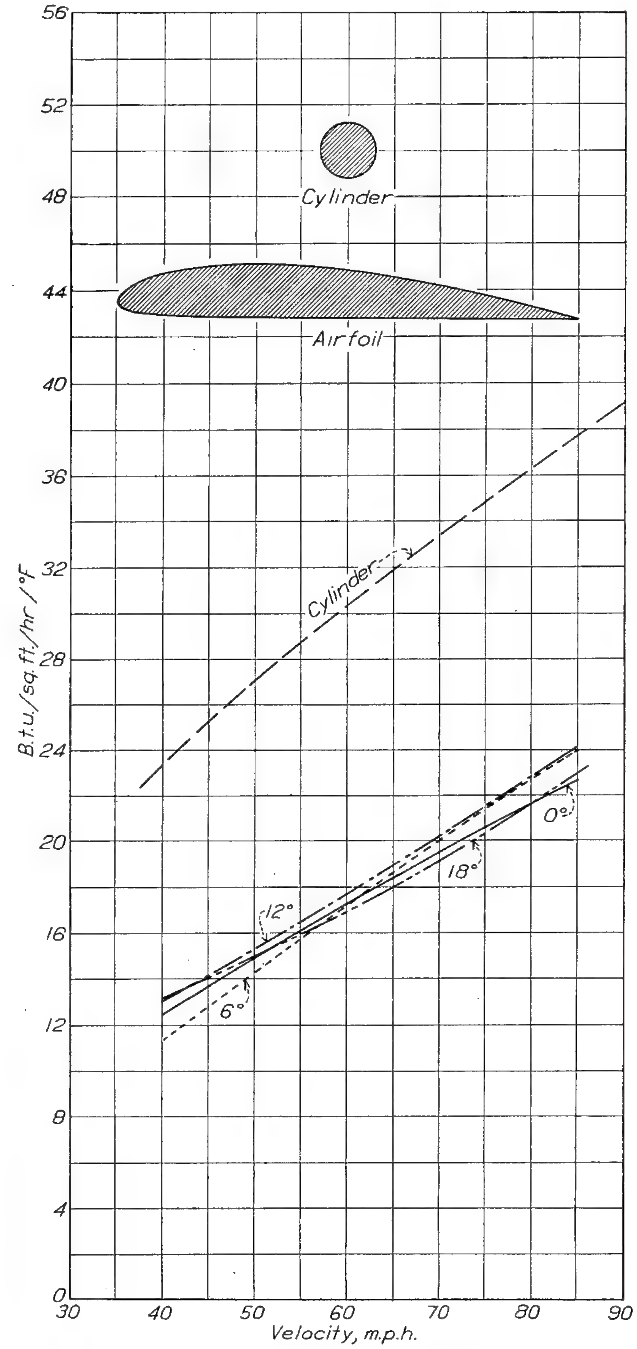

FIGURE 14.—Rate of heat transmission from the entire airfoil and the cylinder

Figures 10 to 13 show a cross plot of the previous curves in which the heat transfer of each section is shown at various angles of attack. The cylinder curve is plotted along with that of section 1 because this

Figure 15 shows curves of the heat transmission at various angles of attack plotted against the chord of the wing section. The plotted points represent the average values obtained for each section plotted in

the middle of each. Hence the curves do not give a very correct conception of the local heat distribution,

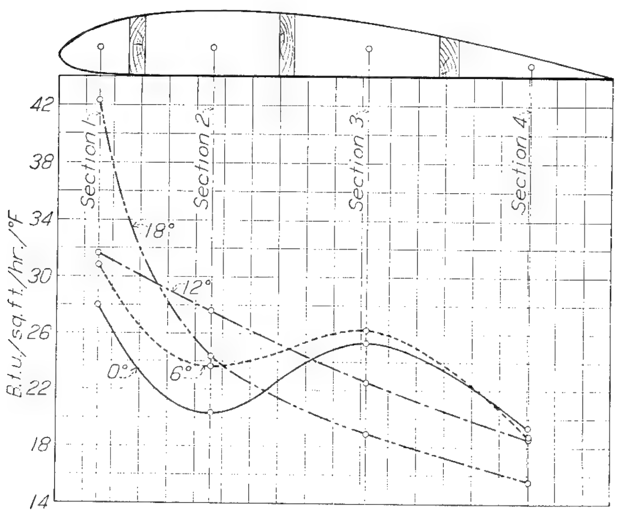

FIGURE 15.—Rate of heat transmission along chord of Clark Y airfoil at various angles of attack. Velocity=85 m. p. h. The curves are based on average values obtained for each section

particularly for section 1, where the relative changes are great. They do, however, indicate the general

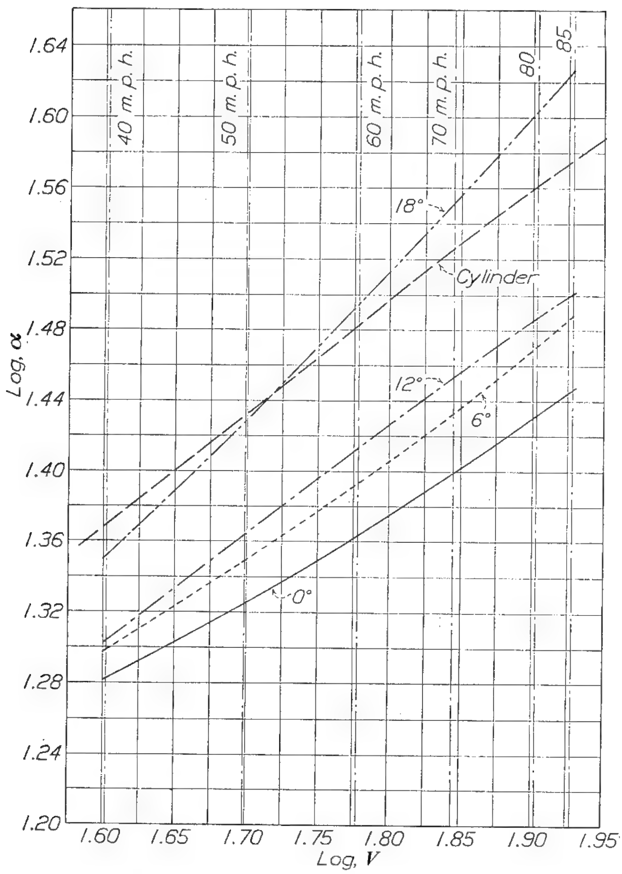

FIGURE 16.—Log α versus log velocity V for section 1 and the cylinder, where α=B. t. u./sq. ft./hr./° F. and V=m. p. h.

change in cooling effects as the angle of attack is increased.

Figures 16 and 17 are curves of heat transfer against velocity plotted logarithmically for section 1, for the cylinder, and for the entire airfoil. These graphs were of assistance in determining the constants c and n defined by the customary relation $\alpha = cV^n$.

Discussion of results.—The tests of the cylindrical tube were included partly to provide a check of the general accuracy of the tests and partly to provide a convenient basis of comparison for the airfoil tests. The curve for the cylinder obtained in Figure 9 is, for the latter reason, replotted in Figures 10, 14, and 16.

In Figure 9 is also shown the curve obtained from formula (I) by Nusselt. This curve lies about 20 per cent below the curve obtained from the pipe tests. In

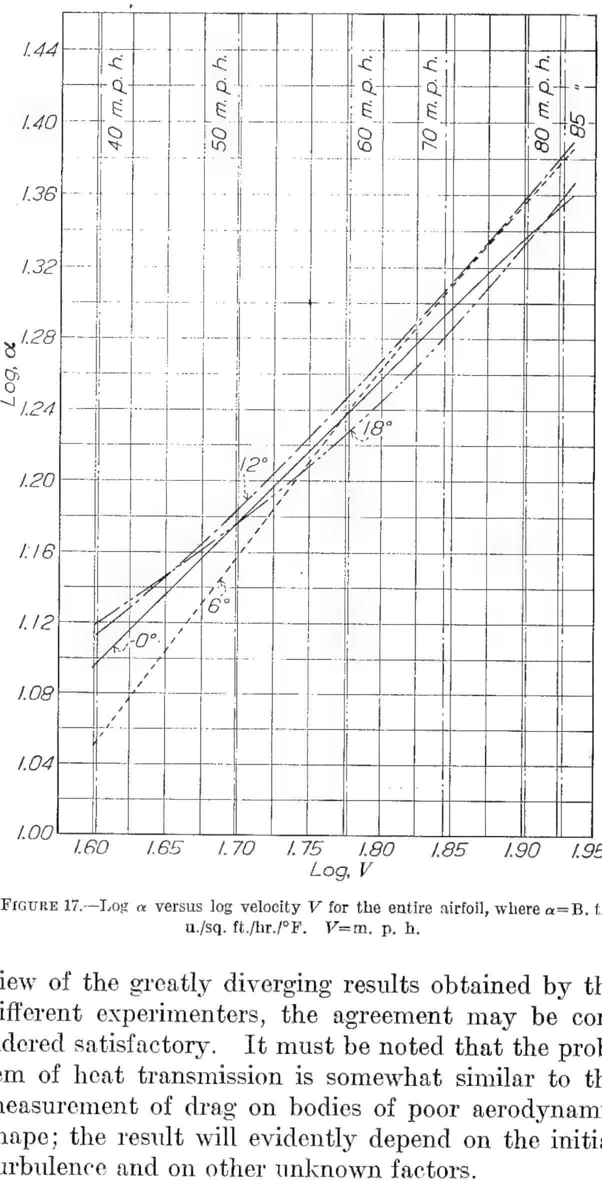

FIGURE 17.—Log α versus log velocity V for the entire airfoil, where α=B. t. u./sq. ft./hr./°F. V=m. p. h.

view of the greatly diverging results obtained by the different experimenters, the agreement may be considered satisfactory. It must be noted that the problem of heat transmission is somewhat similar to the measurement of drag on bodies of poor aerodynamic shape; the result will evidently depend on the initial turbulence and on other unknown factors.

It will be observed from Figure 10 that the heat transmission of the front section (1) at normal angles of attack of the airfoil is only slightly less than that of the cylinder and becomes equal to it at an angle of

attack of approximately 18°. The dimensions of section 1 and of the cylinder are nearly alike.

Figure 14 shows that the heat transmission coefficient for the entire wing at all angles of attack remains at about 60 per cent of the corresponding values for the cylinder. This result is what could be expected in view of the fact that the coefficient increases as the linear dimension of the body decreases. The Nusselt pipe formula indicates that the heat transmission coefficient for large values of the Reynolds Number should be proportional to $d^{-0.284}$. No rational basis exists for a comparison of an airfoil with a pipe. Some idea of the expected change in the heat transmission may be obtained, however, by converting the airfoil to a circular cylinder of the same surface area. The diameter of such a circle is $7d$. The heat transmission coefficient is then

$$7^{-0.284}=0.576.$$

Hence the coefficient of the airfoil is expected to be 0.576 times the coefficient of the cylindrical pipe. The agreement is almost perfect.

Figure 14 shows, as mentioned, that the transmission for the entire airfoil is fairly independent of the angle of attack. This result is very interesting, although not entirely convincing on the basis of a single test series. The results from the British test (reference 2) carried on at lower Reynolds Number show, however, the same tendency. Table IX, page 24, in this reference shows that the value of the heat transmission coefficient for the R. A. F. 26 model at 70 feet per second changes only from 0.805 to 0.856 kilowatt per square foot per 100° C. when the angle of incidence is changed from $-0.9°$ to $+10.4°$.

Regarding the absolute values of the transmission coefficient, however, considerable discrepancy is evident. The British tests on the heat transmission of the R. A. F. 26 show values of the coefficient which are on the whole approximately one-half of the values reported in this paper. Tests by Scott (reference 3), on the other hand, show values which are almost double. Scott points out that the temperature of the leading edge was higher than necessary owing to the fact that only a single heating element located somewhat too close to the leading edge was employed, and that his results for this reason would be high.

The tests mentioned are, of course, not strictly comparable because they were carried on with quite different bodies at different Reynolds Numbers. The discrepancies are, however, too large to be explained on this basis. It is probable that the initial turbulence existing in the tunnels is of great importance. A study of the influence of the initial turbulence on the heat transmission of streamline bodies may disclose valuable facts.

We have shown that the heat transmission coefficient is given by equation (I)

$$\alpha=\frac{\lambda_m}{l}\Psi\left(\frac{lV_o\,\rho_m}{\mu_m}\right)$$

It is commonly agreed that the function Ψ at large Reynolds Numbers, is satisfactorily represented by the form

$$a\left(\frac{lV_o\,\rho_m}{\mu_m}\right)^n \text{ where } a \text{ and } n \text{ are constants.}$$

We may then write

$$\alpha=a\,\frac{\lambda_m}{l}\left(\frac{lV_o\,\rho_m}{\mu_m}\right)^n \qquad \text{(III)}$$

The coefficient n lies somewhere between 1 and 0.5. The results from different experiments are, however greatly at variance as to the exact value. In the present investigation the only variable studied was the velocity V_o.

The values obtained for the constants, c and n, in the equation $\alpha=cV^n$, are given in Table I. These values are obtained by means of the logarithmic plots in Figures 16 and 17. It is observed that the exponent n for the entire airfoil lies in the range 0.73 to 1.0.

The exponent n for section 1, also given in Table I, is much lower, starting at 0.60 for 0° and increasing gradually with the angle to 0.85 at 18°. These values are in perfect agreement with the values given in reference 2 for strip 1 near the leading edge. The fact that the exponent is close to 0.5 indicates that the flow is nearly laminar.

Estimate of heat required to prevent ice formation on full-sized airplane wings.—From Figure 22 it is found that the heat transmission coefficient at 80 miles per hour is 22 B. t. u. per square foot per degree Fahrenheit. The chord of the model wing used is 10 inches. An estimate of the coefficient of heat transmission for an airplane wing having a chord of 7 feet, a span of 50 feet, and operating at a velocity of 80 miles per hour is obtained in accordance with formula (II), using a value of $n=0.85$.

$$\alpha=22\times\left(\frac{84}{10}\right)^{-0.15}=16.0\,\frac{\text{B. t. u.}}{\text{ft.}^2\ {}^\circ\text{F. hr.}}$$

The heat lost by the entire full-sized airfoil at, say, a 10° F. temperature difference is then

$$H=16.0\times7\times50\times2\times10=112{,}000\ \frac{\text{B. t. u.}}{\text{hr.}}$$

or

$$\frac{112{,}000}{2{,}545}=44.0 \text{ horsepower.}$$

Because the model tests were carried on at quite large Reynolds Numbers, the errors involved in applying the results to full scale are quite negligible.

To heat the leading edge only, or rather a region corresponding to section 1, constituting one-seventh of the entire wing surface and having a coefficient of heat transmission which is about 30 per cent greater

than the average for the wing, a quantity of heat equivalent to

$$44 \times \frac{1}{7} \times 1.3 = 8.2 \text{ horsepower}$$

is needed to maintain a temperature of 10° F. above that of the air stream.

It is shown in Part II that the proper *distribution* of this heat is a matter of considerable practical difficulty. The actual quantity of heat is, however, surprisingly small and no mechanical difficulty would be encountered in extracting this quantity from the engine exhaust by any method whatsoever. The heat available in the engine exhaust under the circumstances referred to is about 100 horsepower.

At greater velocities the condition is still more favorable. The heat loss increases with the velocity in 0.5 to 1 power, while the heat available in the exhaust goes up with the third power.

REPORT NO. 403

ICE PREVENTION ON AIRCRAFT BY MEANS OF ENGINE EXHAUST HEAT AND A TECHNICAL STUDY OF HEAT TRANSMISSION FROM A CLARK Y AIRFOIL

PART II

INVESTIGATIONS ON THE DESIGN AND OPERATION OF ICE-PREVENTION EQUIPMENT

Choice of heating system.—Many methods have been proposed for preventing the formation of ice on airplane wings by the use of heat, and among them is the employment of an electrical heating system. The arrangement contemplates the use of an electric generator operated either by power from the propeller engine or from a separate source in order to heat, by an electric current, a system or wires extending over and around the exposed surfaces of the airplane.

The main objection is that power must be generated by means of a gas engine and no engine has a heat efficiency which exceeds 25 per cent. In addition, the efficiency of the generator can not be made to exceed 80 per cent, which means that less than one-fifth of the heat energy of the fuel is utilized, while four-fifths is wasted. The efficiency is thus far below reasonable limits.

For an electrical system to supply an amount of heat comparable to that available from the exhaust, the additional generator engine would have to be of about the same size as the main engine. In any case, the weight of an electrical system for such purposes would be excessive and would represent a too serious loss in pay load to justify its installation.

There are several possible methods for preventing the formation of ice on airplane wings by the utilization of waste heat from the engine:

I. Direct exhaust heat.
II. Hot air heat:
 (*a*) In connection with air-cooled engines with forced air cooling.
 (*b*) With an air heater in the exhaust pipe.
 (*c*) A combination of (*a*) and (*b*).
III. Vapor-heating system:
 (*a*) In connection with a water-cooled engine.
 (*b*) With a boiler in the exhaust pipe.

The first and possibly the most direct method would be to pipe the exhaust gases from the engine directly through the parts to be heated. This method has, however, many serious drawbacks. The acids in the exhaust gases combined with the high temperatures to which such pipes would be subjected would corrode and weaken them and make a frequent renewal of the installation necessary. The danger from fire to an airplane with such an installation would be considerably increased as a slight leakage or a break in the hot piping would be almost certain to start a fire.

In connection with all-metal airplanes, equipped with a detachable leading edge, this method is apparently practicable. The efficient distribution of heat is a matter of some difficulty, but as plenty of heat is available this is of minor importance. Local overheating should be prevented by proper insulation and the exhaust gas should be discharged into the air stream at the wing tips. The arrangement would act as a very good muffler.

The second method is the employment of a hot-air system. It is probable that this system would be very convenient in connection with engines designed to use forced air cooling. The air would be heated by circulating it over the engine and supplied to the wings by a fan circulator. The heat may, of course, also be furnished by an air heater in the exhaust pipe. Air heaters are, however, rather clumsy and heavy and the proper distribution along the leading edge would be difficult.

The combined use of a steam wing-radiator system for cooling the engine and heating the wings presents possibilities for future designs. At present, however, the design of an efficient wing-radiator system which would be satisfactory for both cooling the engine and for preventing ice formation, and which would have adequate provision for protecting the return lines from freezing, has not been developed and is not taken up in this investigation.

The last method mentioned for utilizing waste heat from the engine is the employment of a vapor-heating system which would extract heat from the exhaust gases and distribute it to the wings by the circulation of steam or hot vapor. The rate of heat transmission from vapors is very great and, in addition, the heat is always supplied to the coldest parts so that a tendency exists to keep all parts at the same temperature. In consequence, the temperature required would be low and very efficient distribution of the heat to the airfoil would result. The system could be made light, a suitable liquid could be used which would not corrode the metal parts, and no fire hazard need be involved.

Wind-tunnel test on rain-catch slot.—Large raindrops have a considerable vertical velocity. This

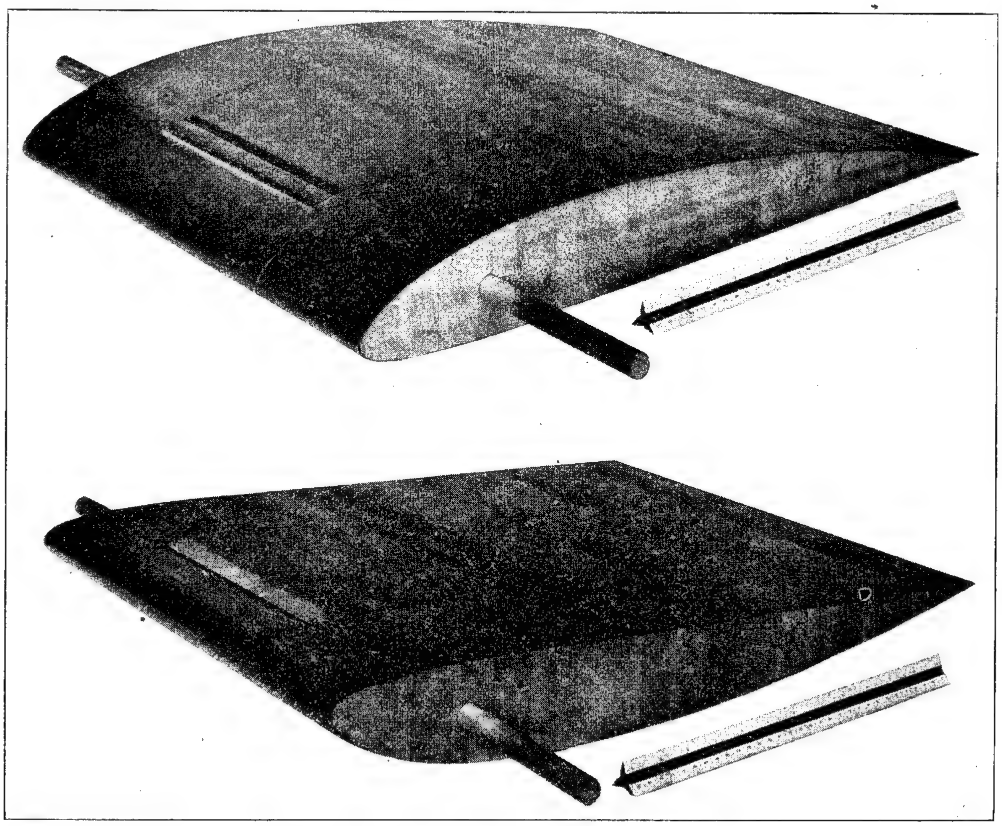

FIGURE 18.—Model for experiments on rain-catch slots

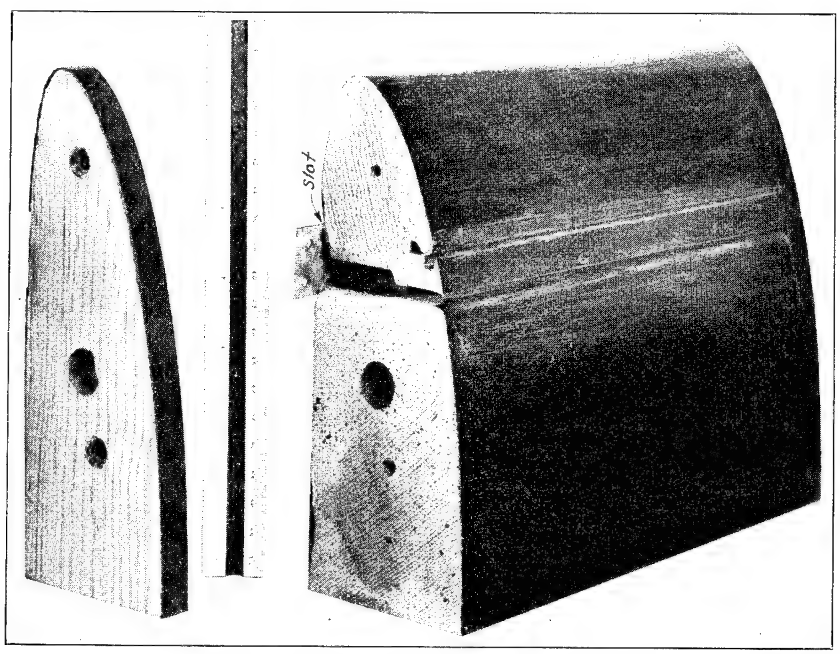

FIGURE 19.—Mid-section of model showing slot construction

velocity reaches about 25 feet per second as a limiting value. The relative angle of attack of the larger raindrops is, at 100 miles per hour, decreased by about 9° as compared with the angle of attack of the air stream. It is then obvious from geometrical considerations that most of the water must strike the airfoil on the front portion of the upper surface, while the remainder of the airfoil, so to speak, is in a "shadow."

Any plan to prevent the formation of ice by heating only the leading edge should include some means to remove the water which falls on the leading edge in order to prevent its blowing back and freezing on the after part of the wing. The most simple and direct method of removing this water appears to be the use of a suitable "slot" located near the leading edge. The following investigation was made on a model to determine the suitable size, location, and effectiveness of such a slot.

A Clark Y wooden airfoil section was made which had a span of 19 inches and a chord of 24 inches. (Fig. 18.) The center section of the leading edge was removable (fig. 19), which permitted the insertion of a variety of leading-edge and slot constructions. The model was mounted in a wind tunnel and a fine spray of water was injected into the air stream in front of it. In this way the effectiveness of the slot as a means of collecting the water which impinged on the leading edge could be observed. Many different shapes and sizes of slots were tested with this model.

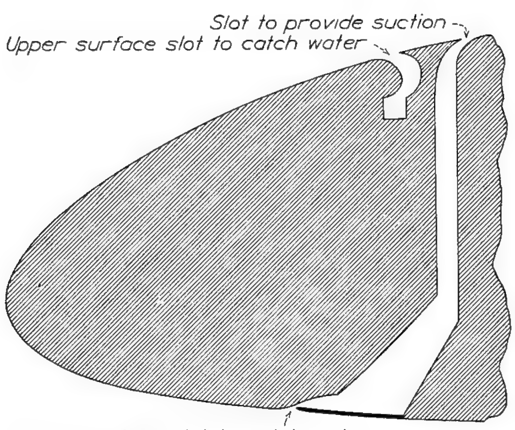

FIGURE 20.—Sketch showing details of model used for experiments on rain-catch lots

The best size and entrance shape for a water slot on the upper surface is shown in the full-sized sketch in Figure 20. The slot leads to a hollow space inside the airfoil which serves to receive the rain water. The actual size of the opening is apparently in no way related to the scale of the model, but is dependent only on the size of the water drops, which is the same on a small model as on a large one.

Several tests were made to determine the effect of the location on the relative efficiency of the slot. Locations more than 10 per cent of the chord length back from the leading edge were found to be quite satisfactory as all the water making contact with the upper surface was completely removed by the slot. At locations closer to the front, the air flow over the surface, being more violent, was found to cause the drops to jump across the slot.

The conclusion is that the slot must be located as far back as is practicable from a design standpoint. The limit is given by the location of the front spar, which is ordinarily located about 12 per cent from the leading edge.

A slot on the lower surface was *a priori* not considered necessary. However, for completeness, a study of such a slot was made.

It was found that no shape of slot was effective in removing the drops from the lower surface unless a means was provided for drawing air into the slot by suction. Of the several arrangements investigated the one most adaptable and effective is shown in Figure 20. The best method for obtaining the desired suction appeared to be the introduction of a suction slot on the upper surface at a point about 13 per cent from the leading edge. The rain slot on the lower surface appeared to operate most successfully at a point about 7 per cent from the leading edge. The operation of the lower slot depends very critically on its design; the opening must be comparatively small, and the front tip of the covering plate must be bent inward slightly.

Figure 18 shows the complete assembly of the test unit equipped with both upper and lower slots, and Figure 19 shows the detail of the center test section separate from the wing blank.

The practical application and construction of a slot on the lower surface to collect the rain water would be very complicated, as it would necessitate the provision of a suction inside the wing for its operation. Such a suction, if obtained by mechanical means, would not only increase the weight of the airplane and introduce an added mechanism to be serviced but would absorb some of the available power from the engine. If the suction were obtained by the employment of an additional slot on the upper surface (fig. 20), the aerodynamic characteristics of the airfoil would be impaired and would result in a lower maximum lift and an increased drag. Hence further investigations were made to determine the practical necessity of using a lower surface slot and to determine whether or not the results of this investigation would be applicable for full-scale design.

Description of apparatus for flight testing.—The principal objects of the following investigation were: (1) To obtain information on the action of a practical vapor-heating system for the prevention of ice formation under full-scale conditions; (2) to evolve practical designs of all necessary ice-preventing equip-

ment; and (3) to confirm tests on the general operation of a rain-catch slot as obtained from small models.

The general problem resolves itself in two parts: (1) The extraction of heat from the exhaust, and (2) its distribution to the leading edge of the wing.

For these tests, a model wing of approximately full-scale dimensions was constructed and mounted on an airplane as an auxiliary wing.

A general arrangement and a diagrammatic sketch of the apparatus used in this investigation is given in Figure 21. A small boiler is shown inserted in the exhaust pipe. The steam passes through the conducting pipe and enters the leading edge of the wing section by means of a steam distributor pipe equipped with small holes as shown. The condensed steam collects at the bottom of the leading edge from which it returns to the boiler through the drain pipe.

A wing model employing a modified Clark Y section was used. A photograph of this wing is given in Figure 22. The profile dimensions of the forward portion are based on a Clark Y section with a 6-foot chord. The section, however, was shortened to an actual chord length of 4 feet. It is obvious that the length or shape of the rear portion is of small consequence in tests of this nature. A 2-foot span of the model was also considered sufficient. The forward or leading edge portion of the wing was constructed separately of thin metal and is shown detached in the lower photograph of Figure 22. Two relief valves, not seen in the photograph, were built into the end of the metal section, one

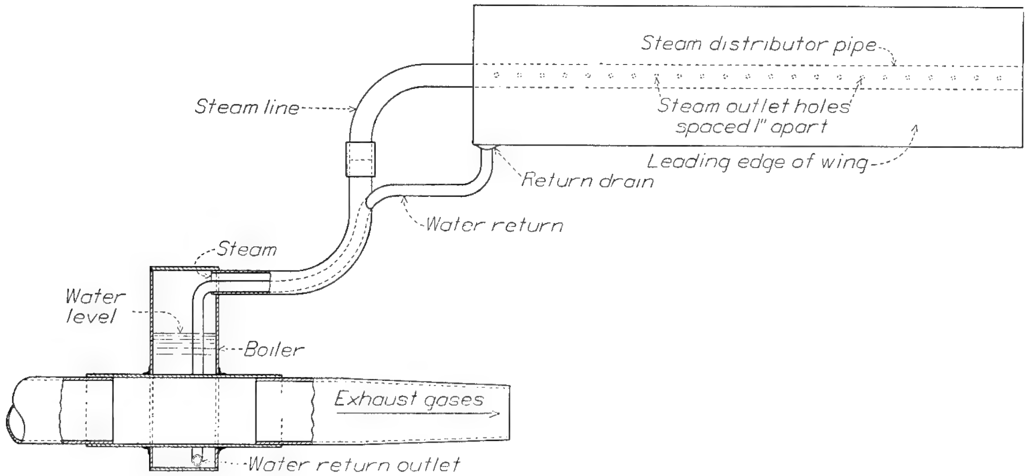

FIGURE 21.—Diagram of vapor-heating system

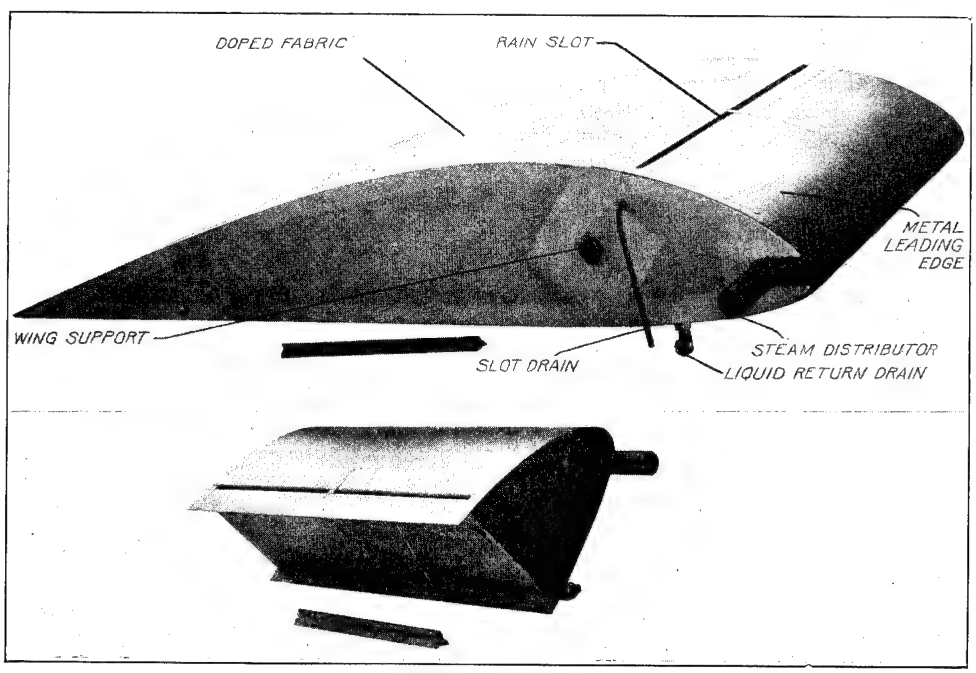

FIGURE 22.—Experimental wing for flight testing

to protect the wing from bursting due to possible excessive steam pressures and one to prevent collapse of the wing from low condensation pressures. The upper slot, as shown in the photograph, merely leads into a separate chamber and is not connected in any way to the steam supply chamber. A small drain pipe leads out from this recess and acts as an exit for the water which is collected by the slot. The slot is located 11 per cent of the chord length back from the leading edge (based on a 6-foot chord). The metal leading-edge section is attached to the wing as shown in the upper photograph of Figure 22. The tube that is shown just aft of the drain pipe is merely a mounting for the support of the model on the airplane.

The boiler is of welded sheet iron equipped with a single 3-inch fire tube and is designed to be inserted directly into the exhaust manifold. (Fig. 23.) The length of the boiler is only 3 inches, and its liquid capacity is about 1 pint.

The model wing and the boiler were mounted on a Fairchild monoplane as shown in the photograph. (Fig. 24.) The boiler and the steam line were carefully insulated. The water return line runs from the drain cock on the lower left-hand corner of the leading edge down into the main steam line and terminates at the bottom of the boiler in the manner indicated by the sketch in Figure 21. A steam pressure gauge facing the cabin is mounted on the upper part of the steam pipe. This gauge was used to assist in checking the operation of the system while in flight.

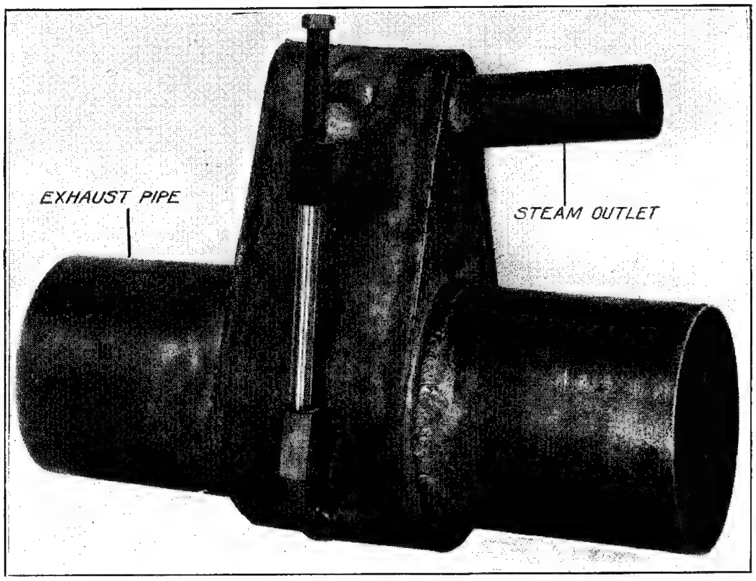

FIGURE 23.—Experimental boiler for flight testing

FIGURE 24.—Experimental installation on airplane

For the purpose of reproducing atmospheric conditions artificially, a water-spraying system was included as part of the equipment. The spraying jets were mounted about 4 feet in front of the wing. The compressed air and water supply necessary for its operation were installed in the cabin.

The first steam distributor pipe installed in the model consisted of a 1-inch pipe extending about two-thirds of the span equipped with a few large side openings placed at random along the pipe. It was found that the steam distribution was very poor, condensation taking place only near the point of entry and along parts of the upper surface.

An excellent distribution was obtained by the distributing pipe shown in Figure 21. This pipe extends all along the span, is closed at the far end, and is equipped with 1/16-inch holes equally spaced along the entire length. The holes are small enough to give the steam considerable velocity and are located so as to direct it onto the particular part of the leading edge where the greatest heat transmission occurs. It was found that the resulting temperature of the entire front section was almost uniform.

No ice formed at any time on the leading edge, and the air temperatures at which tests were made went as low as 18° F. which is well below the range encountered in normal ice storms.

Tests and results.—In choosing the proper liquid to use in the boiler, several things are worthy of consideration. A liquid having a fairly low boiling temperature is to be desired. The total heat withdrawn from the exhaust pipe and distributed to the wing will be nearly constant irrespective of the liquid in the boiler. But the efficient distribution of the heat over the wing surface does depend to a certain extent on the liquid used.

The higher the boiling temperature of the liquid the more difficult is the design of the distributing system. Low temperature vapors tend to distribute themselves evenly over the entire surface while high temperature vapors tend to condense at the very point they strike the cold surface.

The freezing temperature of the liquid is of importance in connection with the possible freezing of the return line in flight and freezing of the boiler when not in use.

The next consideration in the choice of a liquid is its combustibility. The boiler, being in direct contact with the exhaust pipe, renders the use of a combustible liquid quite dangerous in case of a crash or a leak in the boiler.

A fourth desirable but not important characteristic of a boiler liquid is a low specific weight, as an ordinary plane will probably require from 3 to 5 gallons.

The fifth and final consideration concerns the choice of a liquid which will not corrode nor rust the metal parts.

Several different liquids were tested in the apparatus under flight conditions.

The heat distribution obtained with water was so poor in connection with the existing supply system that its further use was not considered. In addition, the return line froze repeatedly.

Alcohol, on the other hand, showed an almost perfect distribution because of its low boiling temperature, but owing to its combustibility the employment of pure alcohol is not recommended.

Carbon tetrachloride was also considered, owing to its noncombustibility and low boiling and freezing points. Although the heat distribution was just as good as that obtained with the alcohol, it was found that it exhibited a strong corrosive action owing to the formation of hydrochloric acid. Owing to this action and also to its excessive weight, the use of carbon-tetrachloride is not recommended.

A further study revealed no other liquid suitable for the purpose, as nearly all liquids having a low boiling point are either corrosive or combustible. Mixtures of water and pure alcohol in a proportion of 2:1 were tried next. The distribution of the heat in the wing was found to be just as good as that obtained with alcohol alone, owing to the fact that only alcohol is evaporated, while the main part of the water remains in the boiler. No freezing of the return line occurred. The fire hazard, due to the alcohol vapor in the main wing, is probably negligible. Tests on this mixture appear to be satisfactory in every respect, and it is recommended for practical use.

Flight tests, made with the rain-catch slot closed, showed that a portion of the water striking the model froze in ridges extending in the direction of the chord from behind the heated section to the trailing edge. It was evident that a considerable portion of the water left the airfoil at the trailing edge without freezing. As all of the water froze at the leading edge in similar tests performed on the unheated wing, the conclusion is that sufficient heat is imparted to the water in passing the heated front section to prevent a portion of it from freezing on the rear part. Part of the water freezes, however, as mentioned, but the formation was found to be rather loosely attached to the airfoil surface and actually blew off as soon as the ridges reached a thickness of about one-half inch. This action was rather surprising, as it is well known that the ice which forms on the leading edge is very firmly attached to the surface. The forces of adhesion are, in fact, usually so great that the only apparent explanation which can be offered to account for the phenomenon is that the rather violent vibrations of the airfoil fabric along the rear section are sufficiently intense to break up the initial thin layer of ice into separate crystals. As the ice builds up on top of this initial layer, the air forces which tend to separate it from the surface become greater. It was found that a temporary

increase in the angle of attack would hasten the removal.

The above-mentioned tests were performed to decide whether or not the slot could be dispensed with. The results indicate that it is quite likely that a slot will not be required on fabric-covered wings. It is probable that a slot will be necessary in connection with an all-metal wing, since the intensity of the vibrations of the covering is much less.

Tests made with the slot open showed conclusively that almost no water passed across the slot. Momentarily a few drops jumped across the gap when the angle of attack of the wing was decreased below that for cruising speed. The water collected by the slot was drained off by the exit provided. In general, it may be said that the slot on the upper surface is entirely effective in eliminating any formation of ice on the afterpart of the wing.

As previously stated, no slot was provided on the lower surface, and in none of these tests were any appreciable formations of ice observed underneath the wing. Only a comparatively small fraction of the total water striking the wing was observed to follow the lower surface.

Unfortunately, owing to the weather conditions prevalent in this section of the country it has in only a few cases been possible to conduct full-scale tests under natural ice-forming conditions. In lieu of such tests, the spray system was employed. The spray produced drops which apparently varied in size from fine fog particles to light raindrops. The time of exposure of the droplets to the cold air was entirely too short to permit the extraction of much heat from the droplets. Consequently, all formations obtained were of the glaze ice type. Figure 25 is a photograph of a formation of ice obtained on the leading edge with the wing cold. The formation appears to be not unlike the glaze ice obtained on wings in natural ice storms. Notice the separate deposits from each of the two jets.

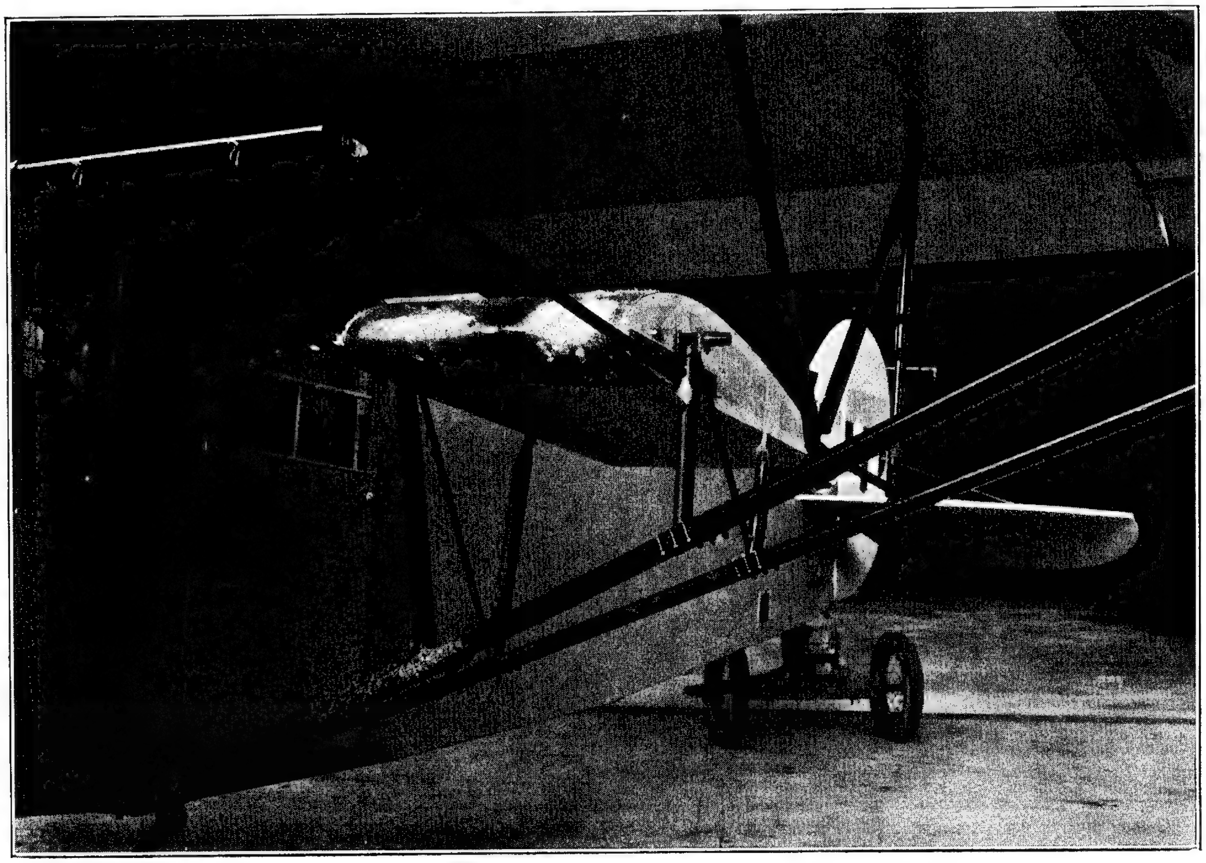

FIGURE 25.—Ice formation obtained with spray on unheated wing

The only way in which the choice of a spraying device might affect any of the results and conclusions derived from these tests would be in connection with the efficiency of the rain-catch slot. Large drops were not obtained with this spray, but by flying the ship in actual rainstorms supplementary information on the effect of larger drops on the efficiency of the slot was secured. Observations on the performance of the slot in natural rain revealed, in general, that the slot efficiency decreases as the drop size increases.

This effect is probably due to the greater relative "angle of attack" of the larger drops with respect to the wing. It is quite possible that some of the larger drops, which have an appreciable downward velocity, strike the wing even aft of the slot.

It was easily observed that greater quantities of water would pass the slot at low angles of attack. In fact, at low angles all the water apparently bridged the slot, while at high angles no water could be seen to cross the gap.

In all tests with the spray which gave small drops, the operation of the slot was almost perfect.

As large drops are seldom encountered in ice storms, it is concluded that the efficiency of the slot will be, in general, satisfactory.

The exhaust pipe boiler used in these tests was of very elementary design and was not intended to be very efficient. An estimate of the efficiency, however, was obtained by measuring the temperature drop of the exhaust pipe, due to the absorption of heat by the liquid. The average efficiency of the boiler was found to be approximately 7½ per cent.

In conclusion, therefore, a boiler with a single fire tube 3 inches long, extracting only 7½ per cent of the available exhaust heat from but four out of the nine cylinders of the engine, prevented successfully the formation of ice on a full-sized leading edge 2 feet in length. It is quite reasonable to expect, therefore, that a waste-gas boiler of ordinary design and of average efficiency would be more than sufficient for the prevention of the formation of ice on the entire leading edge of any full-sized airplane as predicted on the basis of the tests in Part I.

Proposed designs.—The practical design and application of a vapor-heating system for preventing ice formation on airplanes would, of course, depend on the type of airplane and its constructional details. It can not be expected that such equipment can be designed for installation on all the existing types of airplanes, especially not on those equipped with slots or other such devices.

The type of airplane which is particularly adaptable to a vapor-heating system is the high-wing monoplane. With this type the lower surface of the wing generally tapers downwards from the tip to the root; thus a natural drainage of the condensate to the center section is provided. Also, in a high-wing monoplane with the boiler located under the fuselage, the head of water between the wing and the boiler is, in all cases, sufficient to permit the proposed use of gravity feed.

In the case of a biplane the return system would be somewhat more complicated. To create a sufficient flow of steam out of the small holes in the steam distributor pipe for adequate distribution to the leading edge, the pressure in the boiler should be approximately 2 pounds per square inch. This corresponds to a head of water of about 5 feet. If this head is not available, as is the case in the lower wing of a biplane, it would be necessary to install in the return line a special pump

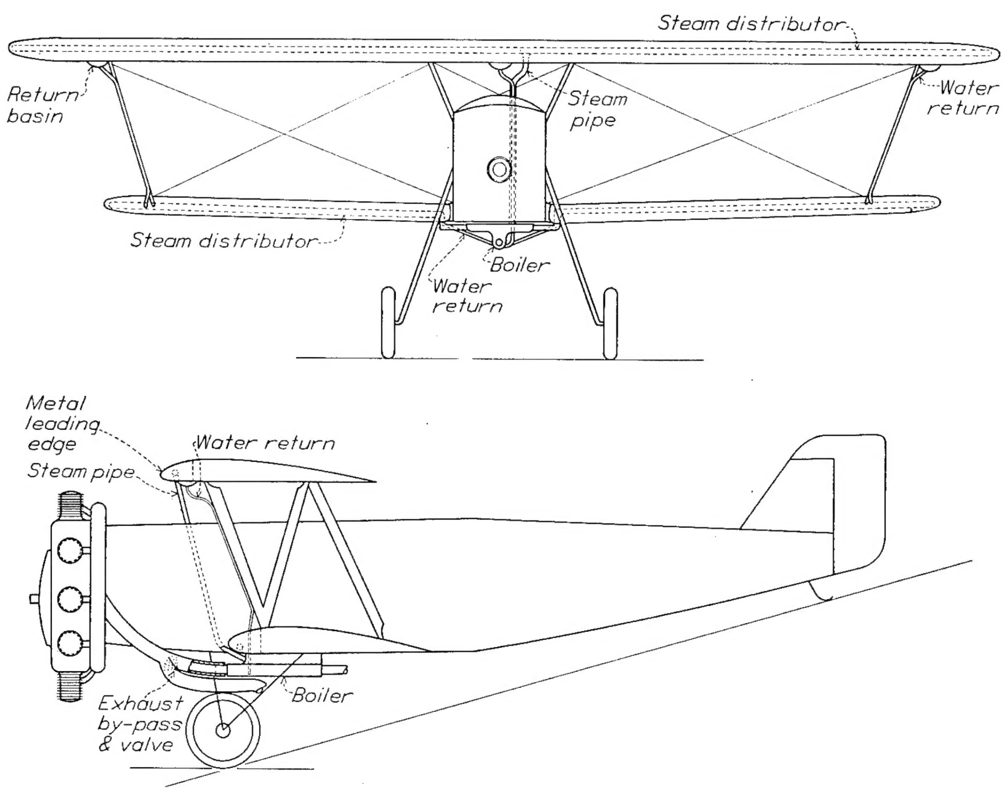

FIGURE 26.—Diagram of suggested layout for biplane installation

to overcome the boiler pressure. The power required would be negligible.

In case the wing system has no dihedral, it would be necessary to provide for drainage of the condensate from several points to prevent the formation of water pockets along the leading edge.

Figure 26 is a diagrammatic sketch showing a suggested layout of a vapor heating system adapted to a biplane. The boiler is shown mounted directly under the fuselage (upper view). The leading edge of each of the three wing panels is connected to the boiler by a separate steam line. As there is no dihedral angle to the upper wing, three water return basins are provided, one in the center and one at each tip. A drain pipe is provided for the center basin which leads directly into the bottom of the boiler. The drain pipes for the wing tip basins, however, are shown attached to the N struts and empty into the leading edge of the lower wings. In this way a large part of the condensate from the upper wing will return to the boiler through the lower wings. Owing to the dihedral

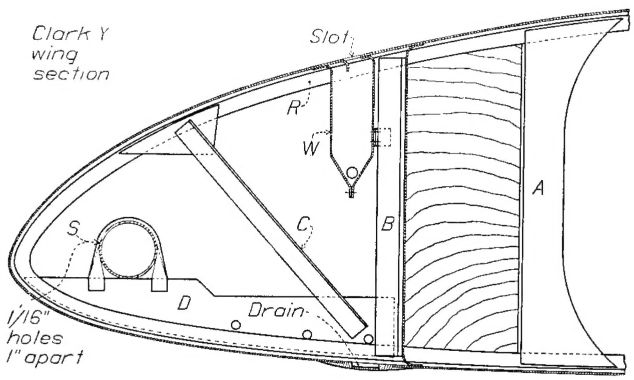

FIGURE 27.—Diagrammatic design of vapor-heated leading edge

angle of the lower wings, the condensate naturally will drain toward the fuselage. The boiler feed pump in the lower wing return line is not shown in the diagram. In the lower diagram (fig. 26) is shown a possible arrangement for by-passing the exhaust around the boiler. The by-pass valve should be operated by the boiler pressure. In this way, the boiler pressure will be controlled, the heat supplied to the wings will be fairly constant, and no presssure relief valve will be necessary.

Figure 27 is a suggestive diagram showing the general leading edge construction of a wing arranged for vapor heating. The construction as shown, with the exception of the rain slot, is not unlike that usually employed for metal wings. The entire construction, including the covering, can be made of duralumin. Referring to the diagram: A, B, C, and R are menbers of the rib structure. D is a light plate which serves several purposes: it stiffens the lower cap strip; prevents any possible surging of the condensate; serves as a gusset plate for the lower junction of members B, C, and R; and supports the steam distributor pipe S. This pipe runs the entire length of the wing and has a series of small holes, in the position shown, to direct the steam onto the leading edge, and is supported and fastened on the plate D by a strap. There may, of course, be several ways to provide drainage of the condensate along the nose, but the drain shown in the figure does not cut into the cap strip, although an inside drain is naturally more satisfactory from an aerodynamic standpoint. A rain-catch slot W is also shown in the diagram, although, as previously discussed, such a slot may not be necessary on all wings. There should be no communication whatever between the space in the slot and that in the leading edge, because the leading edge space must be entirely inclosed to prevent loss of steam or vapor. The slot is interrupted by the cap strip of each rib for structural reasons, but this will not interfere noticeably with its operation. The spar is insulated from contact with the vapor in the leading edge by a thin metal covering which must be sealed at all points to prevent leakage of vapor. The rib members A and B clamp the whole leading edge onto the spar and must be strong enough to withstand all leading edge loads.

CONCLUSIONS

The most essential result obtained from this study is the fact that ample heat is available both in the exhaust and in the cooling water for the purpose of ice prevention.

An analysis of the possible methods for preventing ice formation showed that the employment of a vapor-heating system (or of a direct exhaust-heating system in connection with all-metal airplanes) offers the most convenient and promising solution.

The experiments showed that a vapor-heating system using a mixture of water and alcohol is an entirely practicable method, facilitating, in particular, the correct heat distribution to the wing surface.

It was found that it is sufficient to heat only the front portion to prevent *in fact* the formation of ice on the entire wing.

It was found that the ice, which formed on the rear portion of a cloth-covered wing by water blowing back from the heated leading edge, was attached rather loosely to the surface and did not build up to any appreciable extent before it blew off.

The conclusion is drawn, however, that a small slot may be necessary on the upper surface behind the heated section to collect the rain water, particularly in connection with all-metal wings. The operation and efficiency of such slots have been carefully examined. It was found that the efficiency of the slot in collecting water is decreased greatly if its location is less than one-tenth of the chord length from the leading edge. A slot on the lower surface of the wing was found to be unnecessary.

It is believed that the high-wing monoplane will be the most convenient type for the incorporation of ice-prevention equipment based on the employment of engine waste heat. The successful design of an airplane that will be immune from the dangers of ice accumulation is, as far as can be judged from analysis and laboratory experiments, only a matter of technical development.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
LANGLEY FIELD, VA., *June 12, 1931.*

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TABLE I

CONSTANTS OF HEAT TRANSMISSION AT VARIOUS ANGLES OF ATTACK

Heat transmission $(\alpha) = cV^n$

Values for n			
Angle of attack	Section I	Entire airfoil	Cylinder
0°	0.50	0.79	----------
6°	.58	1.00	----------
12°	.60	.83	----------
18°	.85	.73	----------
----------	----------	----------	0.65
Values for c			
0°	2.96	0.683	----------
6°	2.29	.287	----------
12°	2.22	.598	----------
18°	.98	.879	----------
----------	----------	----------	2.14

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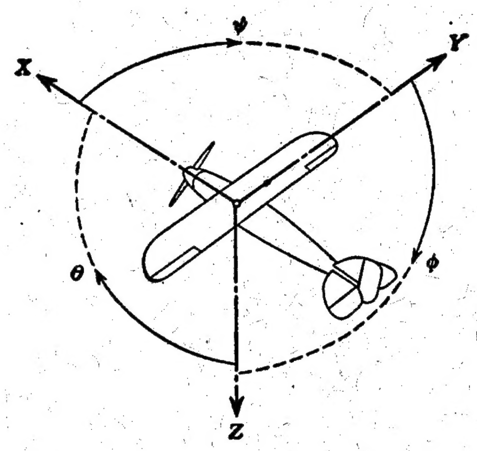

Positive directions of axes and angles (forces and moments) are shown by arrows

Axis		Force (parallel to axis) symbol	Moment about axis			Angle		Velocities	
Designation	Symbol		Designation	Symbol	Positive direction	Designation	Symbol	Linear (component along axis)	Angular
Longitudinal	X	X	rolling	L	$Y \longrightarrow Z$	roll	ϕ	u	p
Lateral	Y	Y	pitching	M	$Z \longrightarrow X$	pitch	θ	v	q
Normal	Z	Z	yawing	N	$X \longrightarrow Y$	yaw	ψ	w	r

Absolute coefficients of moment

$$C_l=\frac{L}{qbS} \qquad C_m=\frac{M}{qcS} \qquad C_n=\frac{N}{qbS}$$

Angle of set of control surface (relative to neutral position), δ. (Indicate surface by proper subscript.)

4. PROPELLER SYMBOLS

D, Diameter.
p, Geometric pitch.
p/D, Pitch ratio.
V', Inflow velocity.
V_s, Slipstream velocity.
T, Thrust, absolute coefficient $C_T=\frac{T}{\rho n^2 D^4}$
Q, Torque, absolute coefficient $C_Q=\frac{Q}{\rho n^2 D^5}$
P, Power, absolute coefficient $C_P=\frac{P}{\rho n^3 D^5}$.
C_S, Speed power coefficient $=\sqrt[5]{\frac{\rho V^5}{P n^2}}$.
η, Efficiency.
n, Revolutions per second, r. p. s.
Φ, Effective helix angle $=\tan^{-1}\left(\frac{V}{2\pi r n}\right)$

5. NUMERICAL RELATIONS

1 hp = 76.04 kg/m/s = 550 lb./ft./sec.
1 kg/m/s = 0.01315 hp
1 mi./hr. = 0.44704 m/s
1 m/s = 2.23693 mi./hr.
1 lb. = 0.4535924277 kg.
1 kg = 2.2046224 lb.
1 mi. = 1609.35 m = 5280 ft.
1 m = 3.2808333 ft.